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MARCH 1971 J. SPACECRAFT VOL. 8, NO. 3

SERT II: Mission, Thruster Performance, and In-Flight Thrust Measurements

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The SERT II spacecraft, containing two 15-cm-diam mercury electron bombardment ion thrusters, was launched February 3, 1970. One thruster operated for 5 months of the 6-month mission goal. The over-all thruster efficiency 0.68 remained constant; the specific impulse was about 4200 sec; and the thrust was 28 mN (6.3 mlb) for a thruster input power of 850 w. A second flight thruster operated 3 months in space. Both thrusters failed due to sudden shorts between high voltages which have since been determined as resulting from the neutralizer location. This paper compares the thrusters performance with corresponding preflight ground testing. The thrust in flight, determined by three different methods, is presented. In addition, the SERT II spacecraft and mission are described.

Introduction

SPACE Electric Rocket Test I (SERT I) in 1964 verified the production of thrust and the neutralization of an ion beam in space.[1] SERT II was launched February 3, 1970, for the purpose of demonstrating 6-month space operation of either one of two ion thruster systems on board. The spacecraft is powered by a nominal 1.5-kw solar cell array.[2] Companion papers describe the auxiliary experiments performed, and the development of the power conditioner.[3–5] The extensive developmental thruster ground testing, which established the confidence necessary for the flight, has been reported.[6–8]

Constant sunlight during the mission was achieved by selecting a polar orbit (Fig. 1) with an altitude (1000 km) and inclination such that the oblateness of the Earth precesses the orbit plane approximately equal to the angular rate at which the Earth moves above the sun. The orbit altitude was an optimization of gravity gradient torques, aerodynamic drag, solar cell degradation, required constant sunlight, and launch vehicle limitations. The Thorad-Agena boost vehicle launched from the Western Test Range (Vandenberg Air Force Base, Calif.), put the satellite into an elliptical transfer trajectory to 1000 km where a second Agena burn circularized the orbit. Once orbit was achieved, the Agena performed several key functions: 1) maintained active control of the satellite while it performed initial steps such as solar cell deployment; 2) mounting platform for the spacecraft and spacecraft support unit (SSU); 3) mounting base for the solar array; 4) gravity gradient orientation structure for entire mission; and 5) provided horizon scanners to determine satellite attitude.

The ion thrusters were offset 10° from the vertical, thus producing enough tangential thrust to raise or lower the orbit depending on the thruster selected. The orbit changing provided a direct measurement of thrust. Real-time thrust was measured by a miniature electrostatic accelerometer (MESA) and calculated from electrical measurements of the ion beam.

The use of flight-proven hardware (except ion thrusters) was an important ground rule. Table 1 lists equipment derived from previous missions. In many cases items were purchased to original specifications; in others, surplus hardware was used. Thermal control of the spacecraft was obtained by entirely passive means using thermal coatings on outer surfaces of the spacecraft and SSU.

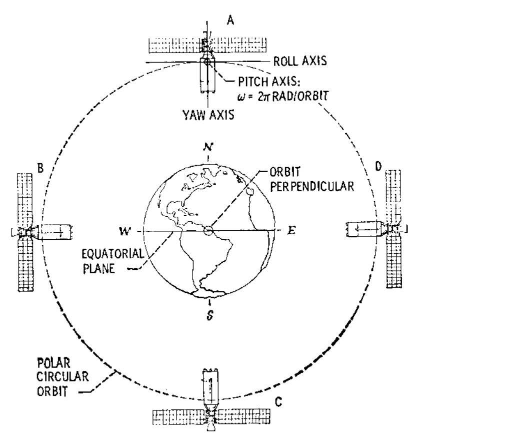

Fig. 1 SERT II circling Earth N-E-S-W orientation.

Presented as Paper 70-1125 at the AIAA 8th Electric Propulsion Conference, Stanford, California, August 31–September 2, 1970; submitted October 1, 1970; revision received December 15, 1970.

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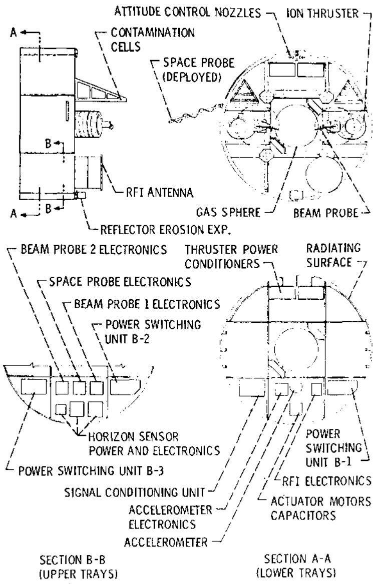

Fig. 2 General arrangement of spacecraft (S/C).

This paper describes the SERT II satellite, presents flight thruster data, and compares it with both prelaunch ground data and prototype thruster lifetesting. The data include electrical parameters which allow evaluation of the thruster power efficiency and durability of various thruster components. The flight thruster main propellant flow was estimated, based upon ground test results, as no means were included in SERT II for direct measurement of main propellant flow.

Satellite Description

The SERT II satellite is 1.53 m in diam, 7.92-m long and weighs 1435 kg. The solar array (193 kg) extends 6 m outward on each side of the Agena aft end equipment rack. The main bulk of the satellite is the empty Agena (740 kg) to which is attached the SSU. The SSU (220 kg) provides spacecraft power conditioning and switching, telemetry and command systems, and the primary attitude control system. The spacecraft (282 kg) supports the ion thrusters and associated experiments.

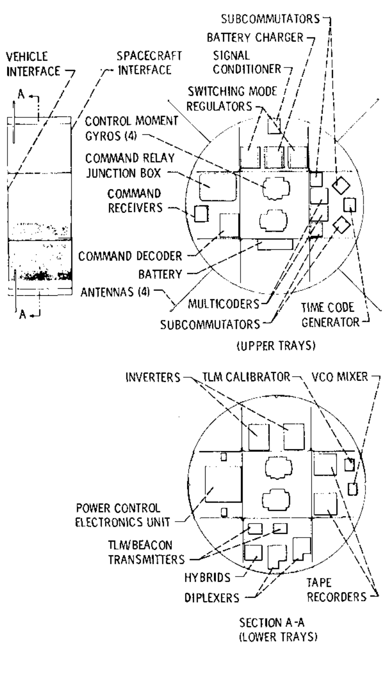

Fig. 3 General arrangement of spacecraft support unit (SSU).

Figure 2 shows the general arrangement of the spacecraft. Each of the completely redundant ion thrusters has its own power conditioning unit. Each thruster is mounted on gimbals so that the thrust vector can be adjusted on orbit through the center of mass of the satellite, thereby minimizing disturbance torques from this source. Several associated experiments are carried: a space probe to measure spacecraft potentials and motor-driven beam probes (described in a companion paper[3]), surface contamination experiments (solar cells held at $+71°C$ and $-43°C$) to determine contaminating efflux from the operating thruster, a radio frequency interference (RFI) experiment to measure any noise generated by the ion thruster in 300–700, 1680–1720, and 2090–2130 MHz bands (these bands will be used by interplanetary and orbiting spacecraft), the previously mentioned MESA, described later. Additional components satisfy the basic requirements for power switching, conditioning and instrumentation. Spacecraft attitude is measured by horizon scanners. A Backup Acquisition System (BACS) is provided for reacquisition should orientation be lost.

The basic shape of the spacecraft is a 149-cm-diam by 53.4-cm-high cylinder. The component access plates form the skin. Three of the skins are aluminum rather than the lighter magnesium because of improved compatibility of aluminum with the Z-93 thermal coating used. Magnesium alloy is used for five outer skins to save weight. The power conditioners (P/C's) are mounted on an aluminum radiator which forms an integral part of the spacecraft structure.

The SSU structure is very similar in construction, size, and appearance to the spacecraft, except for the power conditioning radiator. Seven trays are used in the SSU for conventional component mounting, whereas the spacecraft requires structural adaptation for supporting the large experiments. Three component access skins have a Z-93 paint and five have perforated aluminum tape. The SSU receives power from the solar array, and with its power system, regulates and controls the housekeeping functions and passes power to the spacecraft P/C's. The SSU houses in its center section a control moment gyro (CMG) package which orients the satellite in one axis and provides damping for all axes. Some spacecraft and SSU components are shown in Figs. 3–5.

Moments of inertia are: I_{pitch}, 11,000 kg-m^2; I_{roll}, 8600 kg-m^2; and I_{yaw}, 2750 kg-m^2. Resultant gravity gradient restoring torques for small angles are: 2.21×10^{-5}, 5.25×10^{-5}, and 0.69×10^{-5} kg-m/deg, respectively. These restoring torques maintain the spacecraft orientation stably in conjunction with the CMG's, which provide additional stiffness in the yaw axis and the necessary damping in all three axes. The backup system (BACS) is a cold-gas system which can reorient the spacecraft from disturbance rates up to 5°/sec. Reacquisition is accomplished open-loop, by commands timed and executed by ground control.

The thruster P/C's are mounted on a large radiator which serves as part of the spacecraft structure. The thermal dissipation of the radiator requires that, when both P/C's are off, radiator strip heaters be turned on to maintain an adequate thermal environment.

Power System

Rated at 1500 w of usable power, the solar array is divided into two functional supplies, a 60-v thruster section (1300 w) and a 35-v housekeeping section (200 w). In the stowed position the array is attached to the rectangular truss work aft of the Agena tank section within the envelope of the Agena-Thorad adapter. When deployed it has a length of 6 m on each side of the Agena and a width of 1.525 m, providing an active area of 17.5 m^2. (Detailed design is given in a companion paper.[2]) The 60-v section provides power through motor-driven switches to the P/C system. The 35-v section is fed into switching mode regulators (SMR) in the SSU for regulation and distribution. A block diagram is shown in Fig. 6.

Two SMR's provide a regulated 26.5 v d.c. ± 1%. The main SMR supplies power to the spacecraft loads, inverters, battery charger, telemetry system, and signal conditioning. The standby SMR supplies power to the command system and transmitters. Diode connection of the main SMR to the standby SMR provides power to the command system and transmitters if the standby SMR fails. Command capability exists to use either SMR to supply the spacecraft and SSU loads and also the capability to bypass both SMR's and operate directly from the solar array. Redundant 115-vac, 400-cycle inverters operating from the main SMR output power to any two of the four CMG's in the SSU, to the beam probe actuators and thruster gimbal motors in the spacecraft and to the Agena horizon sensors.

A 40 amp-hr, silver oxide-zinc battery capable of at least five discharge cycles is electrically back biased from the housekeeping solar array and is instantaneously available for emergency conditions. Should the SSU sense an undervoltage condition, the battery comes on line to support essential loads excluding the thruster and experiments. A battery charger is provided to maintain charge status and recharge the battery should it be required.

Power switching for the telemetry and power systems is provided in the SSU, whereas experiment switching is performed in the spacecraft. The command receivers and decoder cannot be switched off. All components in the spacecraft and SSU are fused. Undervoltage protection is provided to disable all systems except the command system and transmitter should the SMR regulated output drop below 23 v d.c. for longer than 200 sec. The P/C's are disconnected from the thruster array should the housekeeping array drop below 23 v for longer than 1 sec.

Communication System

The airborne telemetry system and the supporting ground station and communication network form a composite system which provides the Lewis Research Center (LeRC) Control Center with real-time telemetry and command verification data during periods of ground coverage. Data storage capability via two onboard tape recorders is provided for periods when STADAN coverage is not available.

Data from four subcommutators are fed into a multicoder where the analog-to-digital conversion and time-division multiplexing are accomplished. The pulse-code-modulated (PCM) multicoder output is then fed into a voltage-controlled oscillator (VCO) through a mixer and transmitted at 136 MHz. The over-all modulation scheme is PCM/FM/PM. Timing pulses for experiments and transmission of tape recorder playbacks and command verification are also provided. Redundancy is provided for the multicoder, VCO's, mixer, tape recorder, and transmitter. Although the four subcommutators are not redundant, some key data are on two subcommutators, and data are allocated so that all data from one device are not on one subcommutator.

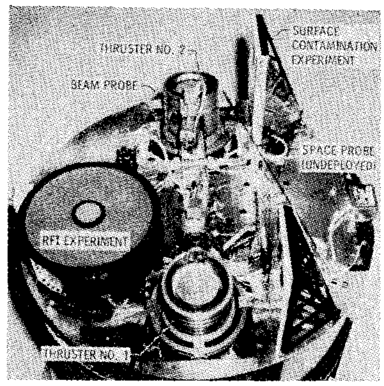

Fig. 5 S/C (bottom view).

The airborne portion of the command system functions in conjunction with the tracking network to achieve real-time command control from LeRC. The command system is an amplitude-modulated, 148-MHz system with a 216-command capability. The spacecraft's receiver demodulates the rf signal and directs the command signal to the decoder, where it is identified, stored, and inserted in the telemetry output. At LeRC the command is verified, and the STADAN operator is instructed to send an execute signal to the spacecraft. Redundancy is provided by use of two command receivers operated in parallel and a command decoder consisting of two fully redundant decoder "halves," each capable of decoding 108 commands. The common telemetry command antenna system comprises a circularly polarized turnstile configuration of four monopole antennas with associated diplexers and hybrids.

Thruster Description

The SERT II thruster[6-9] is a nominal 1-kwe, 15-cm-diam mercury electron-bombardment ion thruster. A detailed description of the thruster system has been previously[9] presented. The propellant reservoirs contain an estimated 8-month (5800-hr) supply of mercury. In operation, liquid mercury is fed by positive displacement (nitrogen gas behind a butyl rubber diaphragm) to a porous tungsten vaporizer plug. The vaporizer plug is heated to pass a controlled amount of Hg vapor into the thruster. Approximately one-third of the thruster flow passes through the thruster cathode. The remainder flows through a flow splitting orifice, into the distributor, then into the main discharge chamber.

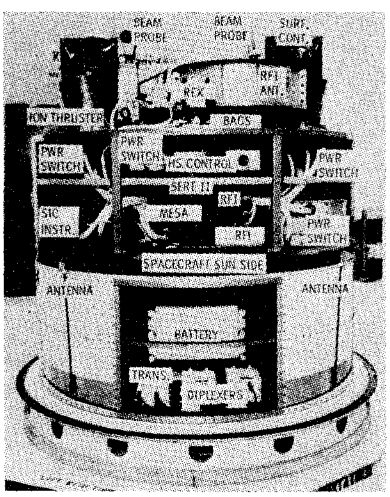

Fig. 4 S/C and SSU (Bay 8).

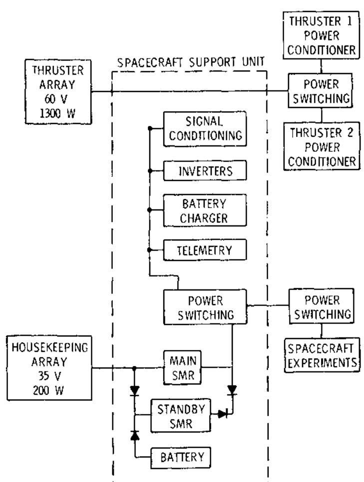

Fig. 6 SERT II power system.

A 1.6 to 2.2 amp discharge is drawn from the hollow cathode to the main anode and, through interaction with the Hg vapor, creates a plasma within the 15-cm-diam discharge chamber. Permanent bar magnets located around the chamber improve the ionization efficiency of the discharge. The baffle and magnetic-field shaping pole pieces produce desirable characteristics of the discharge.[8] Ions diffuse to the screen-accelerator grids, where they are extracted and focused into a 0.25-amp, 3000-v Hg^+ beam. The ion beam is neutralized by an equal current of electrons injected from a hollow cathode neutralizer. The mercury for the neutralizer cathode discharge is supplied by a separate feed system,[7] similar to, but smaller than, the main feed system. A ground screen encloses the entire thruster except for the accelerator grid and prevents ambient plasma electrons from streaming to thruster components at positive high voltage.

Thruster 1 was oriented to raise the spacecraft orbit, and thruster 2 was oriented to lower it. With respect to the thruster life objectives, each thruster system was a backup to the other. The solar array produces only enough power to run one at a time. The solar array output voltage and current are switched directly to the input of the operating P/C. The high-voltage outputs of the P/C are unregulated and vary directly with the solar array output voltage, which may vary with time due to long-term degradation or seasonal sun-array angle changes. Any such variation will cause corresponding changes in specific impulse, thrust, and over-all thruster efficiency.

For SERT II a 2° or greater thrust misalignment will cause spacecraft tumbling. Initially the thrusters were properly aligned, but launch vibration, thermal distortion (most notably that of the grid system), or accelerator grid wear could change the thrust vector direction. To permit realinement, each thruster had a gimbal system comprising an inner ring (an integral part of the thruster structure), an outer ring, two actuators, and pin pullers. The mass of each thruster system fully loaded is distributed as follows: thruster, 3.0 kg; tankage and supports, 4.3 kg; propellant, 15.0 kg (0.9 kg in neutralizer); gimbal system, 7.7 kg; and total, 30.0 kg.

On SERT II, standby thruster (thruster 2) operation was initiated on February 10, 1970. It was operated for two days to confirm correct operation in space and was then shut down. The primary thruster (thruster 1) operation was initiated on February 14, 1970 and it operated to July 22, 1970 (3781 hr), with two brief interruptions. (It was shut down for 17 hr during the solar eclipse of March 7, 1970, and on May 21, 1970, it was shut down for 10 hr because of excessive high voltage cycling.) Thruster 2 was restarted on July 24, 1970 and operated to October 17, 1970 (2011 hr) with one planned shutdown of 50 hr for the solar eclipse of August 31, 1970.

In startup, the thrusters were preheated for a minimum of 1.5 hr to insure that mercury condensation did not occur in any portion of the thruster or feed system. Preheat power was supplied by resistive heating of the main cathode, the propellant isolator, and the neutralizer cathode. The neutralizer keeper discharge was also established during preheat to stabilize its operation and provide a source of electrons when the ion beam was initiated. After preheat, the thruster discharge was initiated without the extraction of an ion beam (high voltages not on). The thruster discharge current was controlled by a feedback loop with the thruster vaporizer. The thruster system then came to near-equilibrium operating temperatures, and thrust could be produced when the high voltages were turned on. Upon extraction of an ion beam, the main vaporizer control automatically switched to the normal control mode that held a constant ion beam current. Stable thrust levels were achieved with either thruster in less than one minute.

The possibility that a badly misaligned thruster would cause the SERT II spacecraft to tumble imposed a mission constraint: thrust initiation and stabilization had to occur while the spacecraft was in direct communication with a ground station. In addition, the thrust level was required to be low enough to allow timely thrust vector correction with the gimbal system if necessary. The total time available while in communication with a given ground station was about 15 min, which dictated that the thrust level be stabilized within a few minutes. The ion beam was first turned on at 40% of full thrust ($\sim$11 mN), then increased to 80% ($\sim$22 mN) in a single step, and then to 100% ($\sim$28 mN). Gimbal adjustment was not required at any thrust level.

The thruster system (thruster and power conditioner) is self-protecting and does not require continuous monitoring from ground tracking stations. Isolated electrical overloading of the high-voltage power supplies causes automatic recycling to clear the overload source. If continuous recycling exists for approximately 100 sec, the thruster system is automatically shut down and must be restarted by ground command.

Ground Tests 1, 2, and 3 and Endurance Tests

Both flight thrusters were operated in a series of three ground tests before flight. Test 1 calibrated the thruster propellant flows (no propellant tanks used) and discharge characteristics over a greater range of thruster operation than expected in flight. Both neutralizer and main flow were measured by timing the fall of mercury in capillary tubes. The power supplies contained flight-like, high-frequency inverter circuits, but with more range flexibility than the flight power conditioner. The test was made in a 1.5-m-diam tank with a stainless steel target 4.5 m from the thruster. The test time was limited to <25 hr to minimize condensed back sputtered target material.

Test 2 verified correct operation of the finally assembled thruster with the flight P/C. Accurate thruster electrical performance data were obtained by reading the thruster system telemetry outputs on a digital voltmeter. This test was made in the same 1.5-m-diam tank as test 1 and was limited to <15 hr thruster operation.

Test 3 was made after the flight thruster had been mounted on the flight spacecraft, and had been subjected to vibration and thermal-vacuum tests as part of the total spacecraft system. This test verified correct thruster system operation on the flight spacecraft and was the last test prior to flight. For this test the LeRC command station was used to initiate commands and to receive telemetered data. The test was conducted in a 4.5-m-diam tank with a stainless steel target located 10 m from the thrusters. Each thruster was operated for $\sim$50 hr. The spacecraft, including the thruster not being tested, was covered with a movable shield to minimize condensed back sputtered material.

Table 1 Components having previous flight history

Components used in SERT II	Vehicle
Horizon scanner, dc-ac inverter, CMB, battery charger, dc-dc converter	Agena
Switching mode regulator and phase-sensitive demodulators	LEM
Battery	Mariner
Command decoder	ISIS
Command receiver, hybrid, and diplexer	Pegasus
Transmitter	FR-1
Tape recorder, solar array	Air Force
Subcommutator, multicoder, frequency division multiplexer, and time code generator	BIOS
BACS	Surveyor or Lunar Orbiter
MESA accelerometer	Saturn

Table 2 Performance of flight thruster 1

Item Index (for tables 2 and 3)

Item	Description	Item	Description
1	Thruster vaporizer heater voltage, V_2, v	12	Neutralizer vaporizer and cathode heater current, I_7, amp
2	Thruster vaporizer heater current, I_2, amp	13	Neutralizer keeper voltage, V_8, v
3	Thruster cathode heater voltage, V_3, v	14	Neutralizer keeper current, I_8, amp
4	Thruster cathode heater current, I_3, amp	15	Neutralizer cathode to tank wall (or to space) voltage, v
5	Discharge voltage, V_4, v	16	Neutralizer emission current, I_9, amp
6	Discharge current, I_4, amp	17	Thruster cathode keeper voltage, V_{10}, v
7	Positive high voltage (screen), V_5, v	18	Thruster cathode keeper current, I_{10}, amp
8	Screen current, I_5, amp	19	Pressure of thruster environment, torr
9	Negative high voltage (accelerator), V_6, v	20	Thruster equivalent neutral flow rate, amp
10	Accelerator current, I_6, ma	21	Neutralizer equivalent neutral flow rate, amp
11	Neutralizer vaporizer and cathode heater voltage, V_7, v	22	Power conditioner input voltage, V_{in}, v

Item	Preheat				Propellant, no beam			30% thrust (11 mN)			
	Test 1[a]	Test 2[b]	Test 3[c]	Flight[d]	Test 2	Test 3	Flight	Test 1	Test 2	Test 3	Flight
1	0	0	0	0	2.67	1.51	1.32	(f)	2.55	1.32	1.17
2	0	0	0	0	1.67	1.63	1.52	1.73	1.73	1.52	1.33
3	>15	15.9	15.7	15.7	8.00	7.20	0.60	[f]4.0	7.50	6.80	7.20
4	2.80	2.88	2.92	2.88	1.43	1.47	1.47	1.50	1.43	1.47	1.47
5	50	>50	>50	>50	39.5	39.6	3.92	38.0	42.3	39.0	38.7
6	0	0	0	0	1.92	1.99	2.13	0.38	0.56	0.59	0.70
7	0	0	0	0	0	0	0	2980	3360	3470	3470
8	0	0	0	0	0	0	0	0.076	0.085	0.088	0.094
9	0	0	0	0	0	0	0	1580	1700	1790	1740
10	0	0	0	0	0	0	0	1.20	0.98	1.03	1.03
11	9.40	8.57	7.53	6.42	8.36	7.12	6.24	7.60	7.42	6.42	5.76
12	2.60	2.30	2.31	2.05	2.20	2.11	1.96	2.20	1.98	1.96	1.78
13	28.2	28.8	29.2	28.6	28.8	29.2	28.6	28.0	28.4	28.6	28.0
14	0.200	0.204	0.241	0.221	0.182	0.213	0.190	0.200	0.184	0.213	0.198
15	0	0	0	-6	0	0	-6	-9	(f)	(f)	-19
16	0	0	0	0	0	0	0	[f]0.075	0.085	(f)	0.094
17	300	>416	>416	>416	8.5	11.3	10.0	14.8	20.1	20.0	18.2
18	0	(h)	(h)	(h)	(h)	(h)	(h)	0.300	(h)	(h)	(h)
19	4×10^{-6}	3×10^{-6}	4×10^{-7}	10^{-10}	3×10^{-6}	4×10^{-7}	10^{-10}	6×10^{-6}	6×10^{-6}	4×10^{-7}	10^{-10}
20	0	0	0	0	(f)	(f)	(f)	(f)	(f)	(f)	(f)
21	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)	(f)
22	[e]60	70	74	70	70	74	68	[e]60	66	67	68

Item	80% thrust (22 mN)				100% thrust (28 mN)							Telemetry uncertainty (rss)
	Test 1	Test 2	Test 3	Flight	Test 1	Test 2	Test 3	Flight 10 hr	Flight 1000 hr	Flight 2500 hr	Flight 3781 hr	
1	(f)	2.55	1.72	1.51	(f)	2.60	1.72	1.51	1.51	1.51	1.51	±0.07
2	1.93	1.93	1.71	1.63	1.96	1.88	1.78	1.71	1.71	1.71	1.71	±0.08
3	[f]4.1	7.50	6.80	6.80	[f]4.4	7.68	6.80	7.20	7.20	7.20	7.20	±0.35
4	1.50	1.43	1.47	1.47	1.50	1.43	1.47	1.47	1.47	1.47	1.47	±0.05
5	38.1	41.5	38.2	38.1	3.70	37.3	37.2	37.1	36.9	36.9	36.9	±0.2
6	1.10	1.05	1.04	1.10	1.68	1.70	1.57	1.67	1.76	1.78	1.80	±0.05
7	2980	2940	3190	3120	2980	2960	3070	3070	2930	2930	2900	±65
8	0.200	0.199	0.203	0.203	0.250	0.253	0.253	0.253	0.253	0.253	0.253	±0.005
9	1580	1540	1640	1590	1560	1530	1540	1540	1490	1490	1490	±50
10	1.30	1.15	1.14	1.25	1.28	1.57	1.49	1.60	1.49	1.60	1.49	±0.12
11	7.33	7.34	6.42	6.00	7.25	7.02	6.68	5.76	6.00	5.76	5.76	±0.25
12	2.15	1.98	2.05	1.91	2.11	1.87	2.01	1.80	1.86	1.80	1.80	±0.05
13	21.9	23.1	23.6	22.9	22.0	23.0	23.6	22.9	22.9	22.9	22.9	±0.7
14	0.200	0.177	0.198	0.193	0.200	0.176	0.193	0.190	0.182	0.185	0.182	±0.004
15	-14.7	(f)	(f)	-19	-17.5	(f)	-14	-27	-24	-27	-36	±2
16	[f]0.196	0.198	(f)	0.205	[f]0.242	0.249	(f)	0.258	0.258	0.258	0.258	±0.006
17	12.0	11.2	12.7	12.3	11.6	10.7	12.3	12.3	11.3	11.3	11.3	±0.5
18	0.300	(h)	(h)	(h)	0.300	(h)	(h)	(h)	(h)	(h)	(h)	(h)
19	5×10^{-6}	5×10^{-6}	4×10^{-7}	10^{-10}	4×10^{-6}	5×10^{-6}	4×10^{-7}	10^{-10}	10^{-10}	10^{-10}	10^{-10}	(h)
20	0.276	[g]0.241	[g]0.265	[g]0.265	0.314	[g]0.309	[g]0.315	[g]0.313	[g]0.313	[g]0.313	[g]0.313	(h)
21	0.027	[g]0.023	[g]0.025	[g]0.026	0.021	[g]0.016	[g]0.017	(f)	(f)	(f)	0.025	±0.002
22	[e]60	60	63	63	[e]60	61	61	61	59	59	59	±1

[a]Calibration test in 1.5 meter diameter vacuum facility.
[b]Test in 1.5 meter diameter vacuum facility, fully assembled, loaded thruster and flight power conditioner; data from digital voltmeter values of hard-lined telemetry.
[c]Final preflight systems test on flight spacecraft in 4.5 meter diameter vacuum facility; data from telemetry.
[d]Flight data.
[e]Laboratory supplies made equivalent to this power conditioning input voltage.
[f]The accuracy of these values is questionable or data unavailable.
[g]Flow rate estimated from ground tests.
[h]No flight telemetry channel.

Prior to the flight, two ground endurance tests were initiated at contractor facilities using thruster systems identical to those on the flight spacecraft. The vacuum facility for each test contains a frozen Hg target located 1.6 m from the thruster. The use of a Hg target greatly reduces the amount of condensed back-sputtered material on the thruster. In

Table 3 Performance of prototype thrusters (item index as in table 2)

Item	Test M, 100% thrust				M, 80% thrust			Test T, 100% thrust				
	Test 1[a]	Life test, hr						Test 1[a]	Life test, hr			
		10	1000	2740	2750	4500	6787		10	1000	3190	5412
1	(c)	[c]1.25	[c]1.25	[c]1.25	[c]1.25	[c]1.25	[c]1.25	(c)	[c]2.40	[c]2.32	[c]1.90	[c]1.89
2	1.86	1.76	1.75	1.75	1.75	1.75	1.75	1.99	1.75	1.72	1.72	1.74
3	7.0	[c]2.9	[c]3.0	[c]3.1	[c]2.7	[c]2.7	[c]2.7	[c]4.3	9.0	9.0	9.0	9.1
4	1.5	1.34	1.35	1.35	1.34	1.35	1.35	1.49	1.40	1.40	1.40	1.41
5	36.9	37.5	37.4	37.3	37.5	37.5	37.5	37.0	36.6	36.4	36.1	36.0
6	1.85	1.90	2.01	2.09	1.31	1.32	1.37	1.62	1.67	1.77	1.95	2.07
7	2920	3180	3190	3190	3230	3240	3240	2990	3130	3130	2920	2840
8	0.250	0.255	0.255	0.255	0.200	0.200	0.200	0.250	0.252	0.253	0.252	0.251
9	1520	1660	1680	1680	1680	1680	1680	1540	1640	1640	1470	1420
10	1.2	1.4	1.2	1.2	1.6	1.6	1.6	1.7	2.6	2.5	2.3	2.2
11	8.1	[c]3.8	[c]3.5	[c]2.6	[c]3.0	[c]2.6	(c)	7.8	4.5	4.4	4.6	4.8
12	2.28	1.98	1.92	1.68	1.76	1.64	(c)	2.17	1.96	1.98	2.00	2.09
13	22.5	22.5	22.3	22.0	22.1	22.0	22±1	22.0	23.0	23.1	23.0	23.0
14	0.205	0.202	0.203	0.200	0.204	0.205	0.205	0.200	0.210	0.212	0.207	0.202
15	-14	-15	-16	-20	-18	-18	-21	-17	-20	-22	-21	-20
16	0.245	0.255	0.255	0.254	0.198	0.198	0.198	247	(c)	(c)	(c)	(c)
17	11.7	11.2	11.3	11.3	12.2	12.6	12.9	12.5	12.5	12.5	(c)	(c)
18	0.30	(c)	(c)	(c)	(c)	(c)	(c)	0.304	(c)	(c)	(c)	(c)
19	2x10-6	8x10-6	2x10-6	2x10-6	1x10-6	1x10-6	1x10-6	3x10-6	2x10-5	1x10-5	1x10-5	5x10-6
20	0.310	(c)	(c)	[c]0.292	(c)	(c)	[c]0.271	0.315	(c)	(c)	(c)	0.318
21	0.023	(c)	(c)	(c)	(c)	(c)	0.014	0.022	(c)	(c)	[b]0.025	0.025
22	60	66	66	66	66	66	66	60	66	66	62	60

[a] Calibration test in 1.5-meter diameter vacuum facility.
[b] Flow rates estimated from previous tests.
[c] Accuracy of data is questionable or data unavailable.

Test M conducted at McDonnell Company (St Louis, Mo.) the P/C was contained in the same tank as the thruster. In Test T conducted at TRW Systems, Inc. (Redondo Beach, Calif.), the P/C was contained in a small, separate vacuum tank.

During the flight and ground Test 3, telemetered data were received in "counts." A range of 0 to 61 counts corresponded to the full range for each measured parameter. The possible error due to this quantizing was combined with all other possible errors (including P/C calibrations) to determine the root-sum-square (rss) error listed in the last column of Table 2. For ground Test 1, 1% meters were used for beam current, discharge voltage, and discharge current; 3% meters were used for the balance of measurements. The propellant mass utilization measurements (using capillary flow tubes) were reproducible to ~1% for the main flow and ~5% for the neutralizer flow.

Table 2 lists the ground and flight performance of thruster 1 under conditions of preheat, propellant (no beam), and 40, 80 and 100% thrust. Table 3 gives performance of Tests M and T, and Table 4 summarizes thruster efficiencies for all tests.

Flight Thruster 1

Preheat

The neutralizer cathode-to-keeper discharge lit properly in all attempts in flight. The time to establish the discharge depends on the thermal time constant of the cathode and vaporizer, and the time required for cathode activation. The time to light the neutralizer was 6 to 12 min in ground tests and was about 4 min in flight. The shorter lighting time in flight was attributable to a higher neutralizer system initial temperature about 55°C in space compared to 25°C for ground tests. A higher thruster system temperature was caused by a combination of factors, which include, a skew sun-spacecraft angle that permitted more solar flux to impinge in the thruster area, slight changes of some thermal control surfaces, and the Agena spacecraft interface which was not present during ground tests. Once lit, the neutralizer cathode discharge was stabilized by its closed-loop control which held a constant neutralizer keeper voltage via a feedback loop with the neutralizer vaporizer heater. The neutralizer vaporizer heater current (item 12, Table 2) was about 0.3 amp lower in flight than in ground tests. This occurred both in preheat and subsequent operation and is probably the result of the hotter thermal environment in space. The Test 1 value, 2.60 amp, was typical of a new start (which requires more neutralizer flow). The Test 1 P/C input voltage was maintained at 60 v, the expected mean value for 100% thrust operation in space.

The next preheat step, initiation of the main discharge with no beam extraction, was accomplished without problems in flight or ground tests. In flight, the main discharge initiated within 0.8 min. Typical ground test discharge initiation times were from 0.5 to 2 min. The thruster cathode power cut back as programed once the main discharge current exceeded 0.3 amp. After lighting, the main discharge stabilized within 1 min at its normal level of 2 amp. Thruster system 1 remained in this propellant mode for 1.6 hr to allow thermal stabilization.

Operation at Reduced Thrust

The flight thruster operation at 40% thrust was similar to that of the ground tests. Propellant flow rates were not measured during Test 1 at 40% thrust because this mode was not planned as a space endurance operating point; the sole purpose was to avoid introducing a spacecraft tumble mode in the event of a major thrust vector misalinement. Other ground tests indicated 50–60% propellant mass utilization efficiency at 40% thrust.[6]

The thruster was at 40% thrust for 3.2 hr during flight. This time was allotted to analyze spacecraft stability and send gimbal adjustments if necessary, but thrust vector alinement was within 1° of the center of mass of the spacecraft, and no gimbal adjustment was required.

The thrust was next increased to the 80% level, and the neutralizer keeper voltage (V_8) set point was changed from 28 to 23 v. The 80% thrust set point was provided to enable the thruster to operate in the event of severe thruster and/or solar cell degradation. The lower V_8 set point was used for endurance running because ground tests showed longer neutralizer lifetimes at the lower voltage.[7] The higher V_8 set point was used for initial operation because of improved neutralizer control stability. The 80% thrust point stabilized within 3 min in flight and was operated for 1.9 hr. The ion beam set point (0.20 amp) and the neutralizer set point (22.9

v), were as expected. Flight operation was in good agreement with ground tests (Table 2).

Operation at Full Thrust

The beam current set point was changed to full thrust (0.25 amp) and the main discharge voltage set point was lowered to 37 v from the 40 v set point. The value of 37 v was chosen to enhance main cathode lifetime.[7] The arrival at full beam in space was accomplished without incident. The thruster stabilized within 2 min at an operating point almost identical to ground tests. The ion beam current was as expected within the accuracy of the telemetry system. An occasional flight reading was recorded at the next higher quantized telemetry value of 0.258 amp.

The level of the main discharge current was of concern, for it strongly influences the main cathode lifetime.[7] The discharge current is sensitive to many parameters, such as ratio of cathode to main flow, screen to accelerator grid spacing, level of beam current, level of beam extraction voltage, strength of magnetic field, and ambient (tank or space) gas density. In flight, the initial discharge current, 1.67 amp, was in excellent agreement with the ground tests; this value rose slowly with time (see the section on Thruster Endurance).

Most other electrical parameters at 100% thrust were similar in space to those measured in ground tests. The thruster and neutralizer vaporizer power required in space were slightly lower, possibly because of the hotter thermal environment on the spacecraft. The screen current (I_5) equaled the neutralizer emission current (I_9) within the accuracy of the measurements. The net ion beam current was equal to the screen current minus the beam current impinging on the accelerator grid. The impinging beam current consisted of both direct and charge exchange ions, but was not the sole contributor to the accelerator current (I_6). Ground tests showed that 0.5 ma of the accelerator current was due to ions created by the neutralizer discharge falling back to the accelerator grid. Therefore, the net ion beam current was computed as ($I_5 \times 10^3 - I_6 + 0.5$) ma.

The level of accelerator current for thruster 1 in space was equal (within data accuracy) to that measured in tank tests. This result indicates that there is negligible charge exchange accelerator impingement produced by interactions of the beam ions with the facility background gas, at the background pressure of 5×10^{-6} torr.

The potential difference between the neutralizer cathode and local ground (tank wall or space plasma potential) is presented in Table 2. For ground tests with thrust being produced the value was -9 v to -18 v, while for space tests it was -19 v to -36 v. This voltage is a function of the neutralizer discharge flow rate, the shape and magnitude of the ion beam, and, during ground tests, the tank geometry. On the SERT II flight this voltage was also dependent on orbital position.[3] This so-called floating voltage represents a loss in ion energy and hence must be considered in calculations of thrust and beam power.

In Table 4 the mass utilization efficiency includes the neutralizer flow. For flight thruster 1, the over-all thruster efficiency in space and ground tests was approximately constant at 0.68. However, the flight power efficiency will decrease with operating time as the positive high voltage decreases (this voltage decrease is directly proportional to solar cell degradation). The estimated flight main propellant flow was based on measured flight electrical parameters and on the ground propellant flow rates obtained for the flight thruster, and other thrusters operated over a similar range of electrical parameters. The average flight neutralizer flow rate, based on propellant tank pressure decay, was 0.025 amp.

On March 7 and August 31, 1970, the SERT II spacecraft twice passed through the moon's penumbra during a solar eclipse. There was insufficient solar flux at these times to supply thruster operating power. The thruster was shut down before the first pass through the eclipse and turned on after the last pass through the eclipse. Shutdown and startup were normal and without incident and all thruster parameters returned to their previous values. Again, no thruster gimbaling was required.

Flight thruster 2 was turned on following the high voltage short of thruster 1. The times to initiate and stabilize the neutralizer and main discharges and beam current were within 0.5 min of those for thruster 1. Ground and flight performance agreed well and the typical operating parameters were similar to those for thruster 1 (Table 2). During the 2011 hr of space operation all thruster 2 parameters (except discharge current) were constant within one count. The change of discharge current is shown in Fig. 7. The calculated performance of thruster 2 is included in Table 4. The calculated thruster efficiency was 0.02 lower than thruster 1. The difference was attributed to both a lower power efficiency because of a higher main discharge current and a lower mass

Table 4 SERT II thruster efficiency and performance summary

Item	Flight thruster 1, 100% thrust							Flight thruster 2, 100% thrust				
	Test 1[a]	Test 2[b]	Test 3[c]	Flight data, telemetry				Test 1	Test 2	Test 3	Flight data, telemetry	
				10 hr	1000 hr	2500 hr	3781 hr				10 hr	2011 hr
eV/ion, discharge	212	215	194	208	222	222	227	227	[d]217	230	250	268
Power eff.	0.89	0.88	0.90	0.89	0.89	0.89	0.88	0.87	0.88	0.87	0.87	0.84
Mass utilization eff.	0.75	0.77	0.76	0.76	0.77	0.76	0.76	0.76	0.76	0.76	0.73	0.73
Thruster eff.	0.67	0.68	0.68	0.68	0.68	0.67	0.67	0.66	0.66	0.66	0.63	0.61
Specific impulse, sec	4090	4180	4200	4190	4150	4100	4070	4140	4110	4180	4000	3860
Thrust, mN	27.8	27.9	28.5	28.5	28.0	27.8	27.6	27.8	[d]27.6	28.4	28.3	27.1

Item	Test M thruster							Test T thruster, 100% thrust				
	Test 1	100% thrust			80% thrust			Test 1	Lifetest data, digital voltmeter			
		Lifetest data, digital voltmeter							10 hr	1000 hr	3190 hr	5412 hr
		10 hr	1000 hr	2740 hr	2750 hr	4500 hr	6787 hr					
eV/ion, discharge	236	243	258	269	208	208	219	203	206	219	244	261
Power eff.	0.88	0.89	0.89	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.87	0.87
Mass utilization eff.	0.75	0.83	0.83	0.83	0.70	0.70	0.70	0.74	0.73	0.74	0.74	0.74
Thruster eff.	0.66	0.74	0.74	0.74	0.62	0.62	0.62	0.65	0.65	0.65	0.64	0.64
Specific impulse, sec	4080	4260	4360	4330	3980	3980	3980	4080	4120	4150	4020	3940
Thrust, mN	27.5	29.3	29.3	29.3	23.2	23.2	23.2	27.8	28.6	28.8	27.6	27.1

[a]Calibration test in 1.5-meter diameter vacuum facility.
[b]Test in 1.5-meter diameter vacuum facility; data from digital voltmeter values of hard-lined telemetry.
[c]Final preflight systems test on flight spacecraft in 4.5-meter diameter vacuum facility; data from telemetry.
[d]Accuracy of data is questionable.

utilization efficiency because of a higher indicated average neutralizer flow rate (0.032 amp).

High-Voltage Breakdowns

The high-voltage breakdowns of the thruster system were monitored throughout the flight. Breakdowns can occur in either the P/C or the thruster. However, it seems more likely that breakdowns will occur in the thruster because of events such as accelerator grids sputtering and building up condensed material which might precipitate a breakdown. A breakdown is defined as a current overload of either the positive or negative high-voltage power supply. If a current overload occurs, both high-voltage supplies are shut down for 0.1 sec and then automatically turned on again.[4] In some cases, the overload clears itself in one shutdown cycle. Other times the overload may cycle several hundred times (10 to 20 sec) before clearing.

The average number of high-voltage electrical breakdowns of each thruster system in space was about the same as experienced in ground Tests M and T. During initial thrusting operation in space, however, there were fewer than expected. During ground tests a large number of breakdowns are usually experienced in the first hour of thruster operation. The time between breakdowns then lengthens over the first 50 hr of operation to a period of about 1 to 10 hr per breakdown. In space, however, there were no electrical breakdowns for the first 8 hr of thruster 1 high-voltage operation. Following this 8 hr there were 38 breakdowns in the next 50 hr. The period between breakdowns then lengthened to an average (for the 3781 hr reported herein) value of about 8 hr. The mean period between breakdowns (for the same 3781 hr) was 6 hr. The maximum period observed between breakdowns was 131 hr, but typically, periods were scattered with 70% of the breakdowns having a period between 3 and 14 hr.

At 2385 hr and without advance indication, thruster 1 sustained high voltage overload cycling for about 100 sec. This amount of recycling caused an automatic shutdown of the thruster. Ground commands were necessary to restart the system. The restart preheat, 30, 80, and 100% thrust steps, were accomplished without incident. The period between breakdowns following the restart was somewhat shorter than typical, there being 20 arcs in the 50-hr period following the restart. Similar sustained overload cycling occurred at 3781 hr on thruster 1 and 2011 hr on thruster 2. In each case the short causing the overload remained, and thruster operation was terminated. Analysis of data during overload operating indicated a short (<10 kohm) existed between V_5 and V_6 (screen and accel grid, respectively). The most probable cause of this short is a fragment of accel grid material which was formed by localized impingement of neutralizer ions (see Thruster Endurance).

Ground Test Thrusters

Ground endurance tests M (McDonnell) and T (TRW) were started in December 1969 with prototype thrusters that incorporated all designs of the flight thrusters. At that time the main vaporizer design was changed. Other components, including the neutralizer system, the accelerator grids, and the power conditioner, however, were reused. New main cathodes were used because they and the main vaporizer were welded together as one assembly. The hours presented for Tests M and T in Table 5 are only those accumulated since the final design change (November 1969) on the prototype thruster. The thruster of Test M and Test T are in every way identical in design to the flight thrusters.

Test M

Prior to the start of Test M all components of the thruster were successfully vibrated at 50% higher levels than expected for launch. The electrical performance was unchanged by the vibration tests and was, in general, the same at the start of endurance testing as it was during Test 1. Minor variations were due to differences in input voltage and power conditioners used. The calculated thruster efficiencies (Table 4) are similar for Test 1 and the life test. At the end of the life test the residual propellant in the neutralizer and main propellant storage tanks was weighed, and the average propellant flow rate determined.

Table 5 Hours of use of components prior to Tests M and T

	Test M, hr	Test T, hr
Thruster cathode-vaporizer	45	44
Neutralizer system	1207	919
Accelerator grid	1207	1534
Thruster body	1207	1534
Power conditioner	1140	1245

Minor interruptions occurred to the life test at 410, 1700, 3620, 4260, and 6020 hr when temporary loss of facility pumping occurred. There was, however, no loss of vacuum, and the life test resumed after a normal preheat cycle.

It is seen from Table 3 that for the life test, all thruster parameters with the exception of the neutralizer heater voltage and current remained nearly constant up to hour 2740. Over this time period the neutralizer vaporizer-cathode heater current showed a decline from 1.98 to 1.68 amp. This type of long-term decline has not been observed in any other endurance test including the flight. At 2740 hr into Test M (3947 hr total on neutralizer system) the neutralizer vaporizer control loop began to oscillate. Although the thruster-neutralizer still functioned, that is, produced a normal steady beam, the thrust level was changed to the 80% value where the neutralizer control loop became stable. The corresponding thruster efficiency and operation at 80% thrust are shown in Tables 4 and 3, respectively. The thruster operated at 80% thrust from 2750 to 6787 hr (main propellant tank empty). The main propellant utilization efficiency estimated from the total mercury used, was 0.08 higher than for Test 1. The reasons for this higher efficiency or the long-term decline in neutralizer vaporizer current are not presently known. The average neutralizer flow for the entire test was 0.014 amp.

Test T

The performance of the thruster during Test 1 and Test T is shown in Table 3. The P/C input voltage was progressively lowered from 66 to 60 v to simulate estimated solar cell degradation. This decrease in input voltage was in part responsible for the variation of the thruster parameters with time and caused the discharge current to increase at a faster rate than in Test M.

The life test was interrupted at 480, 1400, and 2440 hr by temporary facility problems and at 3190 hr to reload the neutralizer reservoir. There was no appreciable loss in vacuum at the shutdowns occurring at 480 and 2440 hr, and the life test was resumed after normal thruster preheat. The facility problem at 1400 hr and the reservoir reloading necessitated exposure of the thruster to the atmosphere. The life test was restarted after cleaning the thruster grids and discharge chamber of a film of condensed sputtered material. Such cleaning was necessary because films with thicknesses of 4 μ or more peel after air exposure. In no test has a film peeled off without removal from vacuum, and this statement includes films up to 40 μ thick.

At the end of the test (5412 hr, contract terminated) both reservoirs were drained. The average flows (Table 3) were near those determined in Test 1. The main flow was averaged over the entire test and the neutralizer flow over the last 2221 hr. The reloading of the neutralizer reservoir was necessary because of 919 hr prior testing at LeRC on a prototype spacecraft. Subsequent analysis showed that neutralizer flows were twice normal during that period due to many restarts and preheat periods.

Thruster Endurance

To the point of failure of each flight thruster, data indicated that the degradation was equal to or less than that indicated in ground life testing. After the failure of flight thruster 2, a number of possible causes were examined. The accelerator grids from Test T (now available for study) showed localized neutralizer ion erosion where several fragments were missing and one fragment that would have soon become detached. With the vertical thruster orientation on the ground, any detached grid fragment would tend to fall out rather than short the grids, and no such short was experienced in any of the ground tests. It was calculated that the gravitational force would exceed the electrostatic attraction for any detached fragment thicker than 150 to 170 μ. Even if the fragments were several times thinner, recent ground tests indicated that the arc welding that occurs in such a short is very weak. Gravity and normal facility vibration could thus remove the shorted fragment for ground tests (but not in zero gravity of flight) during the 0.1 sec off period of the power-conditioner overload cycle.

The presence of gravity, then, invalidated one aspect of the SERT II ground tests. This localized fragment wear can be controlled, however, by repositioning the neutralizer or by providing increased current capability to burn out shorts. Once the shorted fragments were eliminated, accelerator lifetime due to gradual erosion (even including the localized neutralizer ions) would be, as shown previously[6,7] and in tests M and T, no problem for a 6-month mission.

The thruster cathode gradual erosion is indicated by an increase of the discharge current with time (Fig. 7). The overall rate of increase for the flight was less than that of either ground test at 100% thrust. The quantized flight data overlap when the reading alternated between two values. The SERT II useful cathode life was limited by the discharge supply to 2.5 amp. The erosion of the cathode at this limit is small,[7] and if a separate cathode flow control were present, greatly extended lifetime would be possible. For example, the neutralizer discharge flow can be adjusted for minor cathode wear. Neutralizer cathodes have run 7900 hr (Test M) and over 10,800 hr in simulated[7] conditions without failure. Projected lifetimes were at least double the actual hours operated.

The thruster body, anode, and propellant distributor experienced no degradiation. The strength of the permanent magnet field and the flow calibration of vaporizers also were invariant with thruster operating time. All electrical insulators are shadow-shielded against condensed deposits; they showed no measurable electrical leakage. Heater wires and their insulation showed no measurable deterioration in flight or in lifetests M and T.

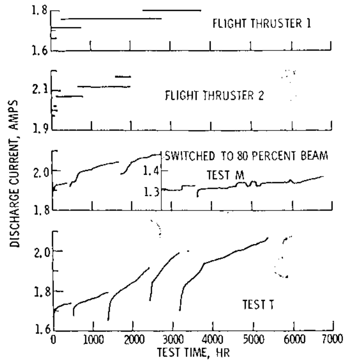

Fig. 7 Trend of discharge current with thruster test time.

In-Flight Thrust Measurements

The thrust of an ion thruster can be most easily, but not necessarily most accurately, determined by measuring the ion beam current and the potential through which the ions are accelerated.[10] These electrical measurements can be readily made in the thruster power supply with little error. Calculation of thrust from these measurements, however, requires corrections for such things as lack of beam collimation and multiply-charged ions.

Direct measurement of thrust is difficult on an ion-propelled vehicle, because the thrust/weight ratio is very small. One can measure the change of flight path caused by the thrust, but very long averaging times are required. The direct measurement of acceleration is better if a sufficiently sensitive instrument exists. The radial component of thrust-induced acceleration on SERT II is $2 \times 10^{-6}\,g$, which must be measured to $\sim$1% error. The MESA (Miniature Electrostatic Accelerometer)[3] theoretically meets these requirements and was flown on SERT II. This section compares results from the three thrust measurement methods: 1) MESA, 2) beam electrical parameters, and 3) orbit radius change.

Thrust Measurement by MESA

The miniature electrostatic accelerometer MESA[11,12] is a single-axis instrument. Electrostatic forces are used both for the support of the acceleration sensing element (the proof mass) and for the measurement of external acceleration. The proof mass is suspended orthogonal to the accelerometer sensitive axis by means of a.c. voltages via series-tuned LC circuits. The magnitude of this voltage is selected to support the proof mass against the expected level of cross-axis acceleration forces. The lower the cross-axis g environment, the lower the required cross-axis voltages. It is advantageous to use support voltages as low as practical, commensurate with the expected environment, to reduce the instrument null bias (output with zero input acceleration), which, in general, is proportional to the cross-axis support forces. The cross-axis suspension capability for the SERT II MESA is 100 μg.

The MESA uses an electrostatic force rebalance method in the sensitive axis to measure acceleration. The frequency of voltage pulses required to maintain the proof mass at its balance point is proportional to the external acceleration. The voltage pulse amplitude, width, and maximum frequency determine the full-scale sensitive axis acceleration capability of the instrument. Full scale is set for 100 μg (even though only about 2 μg are to be measured) to allow accurate calibration on Earth. Unlike most instruments, the MESA measurement error is a percentage of reading, plus null uncertainty, rather than the usual percentage of full scale, plus null uncertainty. Thus, in space the 2% of full scale can be measured with the same accuracy as 100% of full scale except for the null uncertainty.[11] The internal data conditioning provides 100-sec averages of the measured frequency. Because of this feature and large internal damping, the MESA is a steady-state instrument.

The only presently practical method of on-Earth calibration uses accurate small-angle deflection and measurement to produce small input accelerations derived from Earth's g

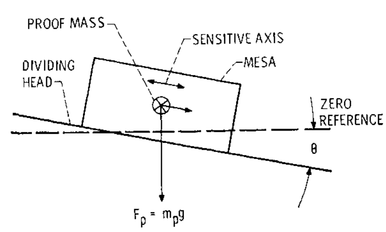

Fig. 8 MESA calibration method.

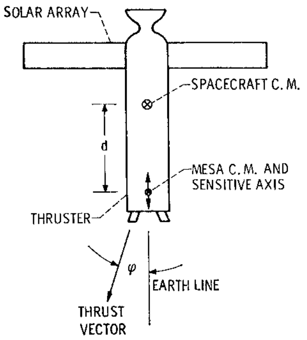

Fig. 9 SERT II acceleration measurement configuration.

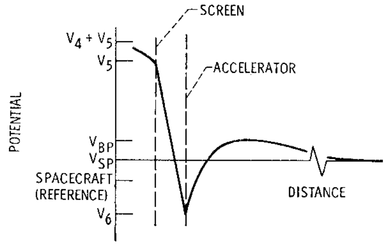

Fig. 10 Thruster potentials.

vector. Figure 8 illustrates this dividing head method. With the sensitive axis perpendicular to the Earth g vector, the mounting surface is varied to obtain an instrument null. The angle θ is then varied to provide input accelerations of g sinθ, or $g\theta$ for the small full-scale angles of 20 arcsec required for SERT II. Use of autocollimator angle measuring techniques for small angles results in measurements accurate to within ± 0.05 arcsec, with a corresponding calibration error of $\pm 0.25\%$. Of course, on-Earth calibration requires a 1-g cross-axis suspension capability for the MESA and results in the accompanying increase in null bias (to 50 μg) due to cross-coupling. Theoretically, the null bias scales down with cross-axis voltages so that the expected value for the SERT II level of 100 μg suspension is 0.005 μg.

In SERT II (Fig. 9), the MESA is located at a distance $d = 2.39 \pm 0.015$ m from the vehicle center of mass (c.m.) with the sensitive axis aligned with the spacecraft yaw axis. The expected sensitive-axis acceleration a (in g's) consists of the thrust-produced term and two simplified orbital components:

$$ag = (F/m_s) \cos\phi + d\omega^2 + 2d\mu/r \tag{1}$$

where F = thrust, N; ϕ = angle between MESA-sensitive axis and thrust vector, deg; m_s = spacecraft mass, kg; ω = orbital rate, rad/sec; μ = universal gravitational constant times mass of Earth, m^3/sec; r = distance from center of earth to spacecraft c.m., m; and $g = 9.807$ m/sec^2. Since the spacecraft is gravity-gradient stabilized (Fig. 1), the MESA revolves about the spacecraft c.m. once per orbit and senses a centripetal acceleration $d\omega^2$. The third term in Eq. (1) is the gravity gradient component, which is due to the MESA being closer to the Earth than the spacecraft c.m.

The orbital accelerations for SERT II were calculated to be 0.73 μg. Other sensitive-axis accelerations due to vehicle attitude and accelerometer misalignment contribute terms two orders of magnitude less than these and are neglected. The orbital acceleration reading of the MESA prior to thruster turn-on can be considered a secondary calibration source for the MESA, although it is impossible to separate null bias from scale factor, since only one data point is available. However, assuming that the scale factor does not change with cross-axis voltage changes, this orbital acceleration measurement can be used to determine the null bias scaling with respect to the 1-g calibration value.

The expected thrust for SERT II was about 28 mN (6.3 mlb). For $\phi = 10°$ and $m_s = 1434$ kg (3162 lb), the thrust-produced acceleration is 1.9 μg. The thrust determined from the MESA acceleration measurements is

$$F = m_s a_T/\cos\phi \tag{2}$$

where a_T = acceleration due to thruster = $a_{\text{on}} - a_{\text{off}}$, and $a_{\text{off}} = a_o + a_{NB}$, where a_o = orbital acceleration, and a_{NB} = null bias. Therefore, the thrust measurement, Eq. (2), is independent of orbital acceleration and null bias (if null bias is constant). The error in F, dF, is a function of errors in m_s, in ϕ, and in the accelerometer:

$$(dF/F)_{m_s} = dm_s/m_s \quad (dF/F)_\phi = \tan\phi d\phi$$
$$(dF/F)_a = da/a \tag{3}$$

where $dm_s/m_s = \pm 0.22\%$,[1] and ϕ is known to within ± 0.5 deg $= \pm 0.0087$ rad. Therefore, for $\phi = 10°$, tan $\phi d\phi = \pm 0.15\%$. The error da/a is composed of several possible errors: 1) scale factor calibration error ($\pm 0.25\%$), 2) scale factor stability and repeatability ($\pm 0.8\%$),§ 3) null bias drift ($\pm 0.5\%$ for 500-sec period), 4) readout system error ($\pm 0.2\%$ for 100 sec; decreases with longer time averaging), and 5) basic instrument linearity ($\pm 0.1\%$ of reading). The root-sum-squared (rss) da/a from these five sources is $\pm 1.00\%$. Therefore, the rss error for thrust (from the errors in m_s, ϕ, and a) is 1.03%. Hereinafter we will consider the error in thrust as determined by the MESA to be $\pm 1\%$.

Thrust from Beam Electrical Parameters

The one-dimensional equation for ideal thrust is [13,14]

$$F = I_B\ [2(m/q)V_{\text{net}}]^{1/2} \tag{4}$$

where I_B = ion beam current, amp; V_{net} = net accelerating potential, v; and m/q = mass-to-charge ratio of a singly ionized propellant molecule, kg/C. Measured values of I_B and V_{net} are telemetered from the spacecraft.

Substituting flight data as received, Eq. (4) becomes

$$F = (I_5 - I_6 + \Delta I_6)(2m/q)^{1/2}(V_5 + V_4 - V_{SP})^{1/2} \tag{5}$$

where I_5 = screen current, amp; I_6 = accelerator current, amp; ΔI_6 = accelerator current due to neutralizer ions, amp; V_5 = positive high voltage, v; V_4 = anode voltage, v; and V_{SP} = space probe voltage, v. The relation of these parameters is shown in Fig. 10, which is a plot of the potential seen by an ion on its way out of the thruster. An error analysis on Eq. (5) yields

$$(dF/F)_i = di/I_5 - I_6 + \Delta I_6 \qquad i = I_5, I_6, \text{ or } \Delta I_6 \tag{6}$$

$$(dF/F)_j = dj/2/V_4 + V_5 - V_{SP} \qquad j = V_4, V_5, \text{ or } V_{SP} \tag{7}$$

The rss error in the SERT II analog flight data is (including the quantizing error of the digital telemetry system) 1.59% of the full scale value of the parameter measured. This was applied to the parameters for each thruster. It was determined that the contributions of I_6, ΔI_6, V_4 and V_{SP} to the error are negligible.

The resulting rss errors for each SERT II thruster at each nominal operating condition during the SERT II mission are given in Table 6. Variations in error and thrust with time arise from two sources: V_5 decreases as the solar array degrades; V_4 decreases as the thruster wears with operating time.

Not all the propellant in the SERT II thruster is ionized; approximately 15% is expelled as neutral particles.[15] The net momentum contribution of this efflux to the total thrust is negligible (<0.001%).

Some of the mercury can be doubly or triply ionized. Studies[16,17] have shown that, for nominal SERT II thruster

§ This much variation was found between calibration tests following the vibration, shock, and thermal-vacuum qualifications tests. (It is possible that other calibration errors, such as test base stability, are contributing to this variation rather than actual physical changes in the MESA.)

Table 6 Errors determined by Eq. (5)

Nominal thrust	Error, rss, % Thruster 1	Thruster 2
30	5.5	5.7
80	2.7	2.6
100 (10 hr)[a]	2.2	2.2
100 (100 hr)[a]	2.3	
100 (3781 hr)[a]	2.3[b]	

[a] Time from thruster beam turn-on.
[b] Estimated from projected operation.

operating conditions, a reduction in thrust of 0 to 2.5% can be applied. We will assume 0% reduction.

The problem of unwanted thrust vector deflection $d\phi$ due to grid system misalignments has been treated using a digital computer program.[18] For SERT II, $d\phi$ due to translational misalignment can be 1.46° or less; to skew, 0.04° or less; and to rotational misalignment, negligible. Each SERT II thruster gimbal system is capable of moving ϕ through $\pm 10°$, in two orthogonal axes, from the preflight geometric alignment through the spacecraft c.m. The anticipated in-flight corrections to ϕ have been unnecessary for SERT II. The attitude data show $d\phi \leq 0.63°$ for each thruster under all operating conditions. (In addition, ground testing of a flight-configuration SERT II thruster showed $d\phi \leq 0.5°$). Since cos0.63° corresponds to a thrust reduction of 0.002%, this will be neglected.

Finally, the loss in thrust due to divergence of the ion beam and the ion current density distribution has not been determined for SERT II, but experience leads to an estimate of a 1 to 3% for the combination of these effects;[19] we shall use 1%.

In summary, the thrust, as determined by Eq. (5), using SERT II thruster flight operating data, is reduced by 1% due to ion beam divergence and in addition is subject to the rss errors of Table 6.

Thrust from Orbit Radius Change

By solving the equations of motion and integrating over the averaging time, we obtain

$$F = m_s\mu^{1/2}(r_o^{-1/2} - r^{-1/2})/t \sin\phi \qquad (8)$$

where t = time, sec; and r_o = orbit radius at time $t = 0$, m ($\sim 7.4 \times 10^6$ m). With the expected thrust level of 28 mN, r will increase by about 570 m/day. An error analysis shows that the individual errors to be root-sum-squared to obtain the error in F are dm_s/m_s, dt/t, ctn$\phi d\phi$, and $2^{1/2}dr/\Delta r$, where $\Delta r = r - r_o$. As was stated before, $dm_s/m_s = 0.22\%$. For reasonably long averaging time (days), dt/t is negligible. For $\phi = 10°$, and $d\phi = 0.5°$, ctn$\phi d\phi = 5\%$. Since $\Delta r/\Delta t$ = 570 m/day, the last error term becomes $2^{1/2}\, dr/570\, T$, where T is the time in days. Since the stated dr is ± 300 m, this term (in %) becomes $\sim 75/T$, and $dF/F = [(0.22)^2 + 5^2 + (75/T)^2]^{1/2}$, which drops from ∞ at $T = 0$, to 5.8% at T = 25 days, to 5.0% at $T = \infty$.

Table 7 Comparative thrust measurements for thruster 1 of SERT II

Method	Thrust (mN) at three nominal levels, mlbs 30%	80%	100%	Error band
MESA	10.4	22.7	27.4	±1%
Electrical				
Uncorrected	11.3	23.2	28.5	±2.2%
Corrected	11.2	23.0	28.2	±2.2%
Orbit change	N/A	N/A	28.0	±5%

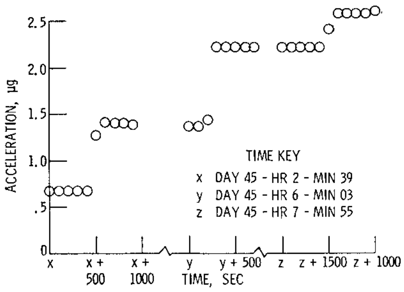

Fig. 11 MESA acceleration data during thruster turn on.

Thrust Measurement Results and Discussion

Table 7 shows the measurement results for each phase of the thruster 1 startup operation. The MESA data were treated using two different averaging techniques. One used only that data obtained within 500 sec before and after each step change in thrust. Figure 11 shows the MESA data edited and plotted to show only these step changes. The step changes in acceleration were then summed to obtain the 80% and 100% thrust levels. The a_T for the on conditions is: $0.725 + 0.866 + 0.328 = 1.919$ μg. The second averaging technique involved using all of the MESA acceleration data available at the time of writing. The difference between the average 100% a_{on} (2.60 μg), and the average a_{off} (0.67 μg) gives an a_T of 1.93 μg, in close (0.5%) agreement with the step-change data.

Figure 12 illustrates typical MESA data for two full orbits and shows the as yet unexplained orbital variation pattern observed in much of the MESA data, whether the thruster was operating or not. Computer studies are being made to correlate these 0.1 μg transient peaks with spacecraft location over the Earth. The close agreement between thrust calculations based on short-term and long-term averaged data supports the contention that these variations are not significantly related to the MESA thrust-measurement accuracy.

Information on a_{NB} was obtained from the MESA readings prior to thruster turn-on. As described earlier, the expected a_o for SERT II was 0.73 μg. This a_o discrepancy can be attributed to an a_{NB} of -0.06 μg, but, as noted just prior to Eq. (1), a_{NB} was expected to scale down to -0.005 μg. The null bias of an accelerometer should either be insignificantly small, or known and stable. For SERT II, an a_{NB} of 0.06 μg is greater than 1% of the measured a_o and hence, is not negligible. Although the long-term drift of a_{NB} appears to be insignificant on the basis of available data, the aforementioned orbital variation of 0.1 μg presents problems in evaluating the short-term null bias stability. On the basis of a_{off} data, the total acceleration variation for periods of less than 500 sec was generally less than 0.01 μg. On the basis of these data, therefore, a_{NB} was sufficiently stable to result in measurement errors of less than 0.5%.

The data obtained from beam electrical parameters are more complete than those from the MESA or orbit-changing techniques. The MESA was not operating during thruster 2 operation and also ceased functioning properly about four

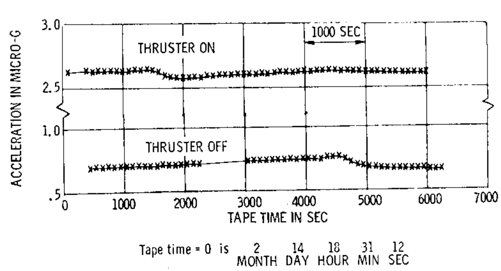

Fig. 12 MESA acceleration data.

days after thruster 1 came on. Thus, the MESA data for thruster 1 do not apply for data after the 100% thrust, 10-hr data. The orbit-changing technique requires too long an averaging period and exhibits too large an error band to list data for all of the thruster operating levels.

In conclusion, the thrust of the SERT II thrusters has been measured to within $\pm 1\%$ by the MESA, to within $\pm 2.2\%$ by the electrical parameter method, and to within $\pm 5\%$ by the orbit-changing method.

Concluding Remarks

SERT II has provided long operational experience with solar-powered space ion thrusters. Both thrusters in general performed the same in space as in ground tests. Thruster 1 ran for 3781 hr and thruster 2 for 2011 hr. Each thruster ceased operating suddenly due to a high voltage short most probably caused by a loose fragment of accelerator grid material. All other thruster lifetime trends were equal to or better than those for ground tests, and if this fragment were eliminated, the thrusters should operate until their propellant supply is exhausted, approximately 6000 hr. (These fragments did not cause shorts in ground tests because they could fall away.) Two ways exist to eliminate future fragments: 1) move the position of the neutralizer to stop localized ion impingement that produces the fragments, and 2) provide sufficient current capability to vaporize shorted fragments.

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